Implementation of an Autonomous Multi-Maneuver Targeting Sequence for Lunar Trans-Earth Injection

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Using a fully analytic initial guess estimate as a first iterate, a targeting procedure that constructs a flyable burn maneuver sequence to transfer a spacecraft from any closed Moon orbit to a desired Earth entry state is developed and implemented. The algorithm is built to support the need for an anytime abort capability for Orion.

I. Introduction

BASED on project requirements, the Orion spacecraft must be able to autonomously calculate the translational maneuver targets for an entire Lunar mission. Translational maneuver target sequences for the Orion spacecraft include Lunar Orbit Insertion (LOI), Trans-Earth Injection (TEI), and Trajectory Correction Maneuvers (TCMs). This onboard capability is generally assumed to be supplemental to redundant ground computation in nominal mission operations and considered as a viable alternative primarily in loss of communications contingencies.

Of these maneuvers, the ability to accurately and consistently establish a flyable 3-burn TEI target sequence is especially critical. The TEI is the sole means by which the crew can successfully return from the Moon to a narrowly banded Earth Entry Interface (EI) state. This is made even more critical by the desire for global access on the lunar surface. Currently, the designed propellant load is based on fully optimized TEI solutions for the worst case geometries associated with the accepted range of epochs and landing sites. This presents two challenges for an autonomous algorithm: in addition to being feasible, the targets must include burn sequences that do not exceed the anticipated propellant load.

II. Algorithm Design

In this paper, an adaptive targeting algorithm that merges an initial guess procedure with a robust iterative non-linear solver is presented. This algorithm has flexible application to either ground based targeting and/or onboard computation as it is designed to converge on solutions quickly and robustly. Since a fully integrated solution is required that handles third body effects which cannot be solved for analytically, a patched conic method has been developed to establish the initial guess.[1] At this point an iterative scheme that minimizes the discrepancy in the states of the trajectory is necessary.

The Trans-Earth Injection (TEI) sequence has a general design of three consecutive burns due to the significant cost savings that exists over a single burn in the full range of anticipated mission scenarios. While both single and three burn solutions can be obtained, the total cost of a three burn sequence is almost always less expensive. This is especially true when the geometry requires a large plane change from the initial Moon orbit to the Earth-Moon transfer arc and is used to determine propellant loads. Regardless of the date or orbit, the three burn sequence has the same basic structure. The first burn significantly raises the apolune of the Low Lunar Orbit (LLO). This is generally the largest maneuver of the three conducted in the plane of the initial Moon orbit. The second burn occurs at or near to the apolune of this new large elliptical orbit and places the spacecraft in the departure plane. The cost to change the direction of the velocity vector is cheapest when the velocity magnitude is at a minimum. This is generally the least expensive maneuver. The third and final burn occurs when the spacecraft sweeps back towards the Moon near perilune. This final burn places the spacecraft on a very specific trajectory designed to hit a particular entry interface (EI) condition that can, depending on the landing site, have very tight constraints. Parameters involved in the selection of a viable EI state include altitude, flight path angle, azimuth, longitude, and latitude targets.

Flexibility to accurately set up targets for the full range of epochs and orbits autonomously is an important attribute of the algorithm. To that end, a vector based approach is used that purposely avoids the use of complex trigonometric expressions due to their intrinsic singularities. As such, vector equations have been derived to compute an associated v_∞ vector for an Earth return trajectory analytically. This vector is determined by the desired

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flight time and calculated from the resultant transfer arc between the current Moon state and associated antipode state at the Earth.[1]

Targeting a v_∞ vector is different from successfully reaching a specific EI state. Over a multiple day transfer the impact of Earth and Sun gravitational perturbations can invoke large changes in the final state at the Earth leading the spacecraft far from the desired EI targets. But to correct these discrepancies, instead of propagating forward to EI, the optimization parameters associated with the target Earth state are propagated backwards in time. Setting a single point to match position and velocity for a continuous trajectory in the Moon sphere of influence (SOI) reduces the convergence difficulties associated with the large swings in EI parameters that correspond to small changes near the Moon.

The desire for a flexible autonomous algorithm is naturally coupled with a need to minimize excessive iteration. Once the analytic guess is determined, it is vital that the targeter quickly converge to a feasible, flyable solution. There exists a variety of algorithms that take advantage of the assumption that the solution lies within the vicinity of the initial guess by designing search directions for the variable control parameters that point towards the minimum feasible solution as efficiently as possible. The TEI autonomous targeter takes advantage of two of these techniques. One option is a full optimization mode, where the constraints and total fuel consumption objective function are simultaneously minimized. A Sequential Quadratic Program (SQP) is selected as the algorithm for full optimization due to the quadratic nature of the solution space where the initial guess is in the vicinity of the solution. This option requires more iteration than the second option, which is a minimum norm mode. The minimum norm algorithm does not attempt to minimize the solution but instead chooses a search direction based that minimizes the constraint error. This correction is calculated using the simple pseudo-inverse mathematical operation. Without a Δv constraint, this approach always converges robustly close to the fuel cost of the initial guess. If a mission cost constraint is added in the form of a maximum quantity of fuel (Δv limit), convergence is still robust, but careful attention must be made to the constraint value used. The optimal value in a given Epoch and geometry is not known apriori and if the limit is too constraining the algorithm could be set up to solve a problem in which no solution exists. As long as the propellant load exceeds performance estimates, this value could be set to the propellant limit, assuming margin.

Once a converged impulsive solution has been obtained, the targeting algorithm can convert each individual Δv into a finite burn command. A two point boundary value problem is setup where the number of controls and constraints are equal. Using a velocity direction based spherical formulation of the burn a simple non-linear solver may be used. The minimum norm algorithm converges these finite burn conversions robustly, even for low thrust engines.

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References

[1]Saudemont, R. R. and Cesar A. Ocampo. "Initial Trajectory Model for a Multi-Maneuver Moon to Earth Abort Sequence," *AAS Space Flight Mechanics Meeting*, 2009.